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NAVAL POSTGRADUATE SCHOOL

Monterey, California

THESIS

(6) AN EXPERIMENTAL STUDY TO DETERMINE THE REDUCTION IN ULTIMATE BENDING MOMENT OF A COMPOSITE PLATE DUE TO AN INTERNAL DELAMINATION.

by

(10) Robert Gary Sprigg

(9) Master's thesis,

(11) Dec~~ember 1~~977

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Thesis Advisor: R. E. Ball

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The purpose of this study was to determine experimentally the effects of internal delaminations in a graphite-epoxy composite plate on the plate's ultimate bending moment. The experiments were conducted using 4-inch by 7-inch specimens with a balanced 0, ±45, 90, 8-ply layup. The delaminations were created by inserting a thin teflon disc between two lamina during layup preparation. The location of the disc, i.e. delamination, was varied in each test, and two disc sizes were considered. The test results revealed that delaminations located near an outer surface resulted in a greater reduction

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in the ultimate moment than those located near the center of the layup. Furthermore, the reduction in ultimate bending moment was found to be independent of the disc size. The tendency for the internal delaminations to propagate at relatively low load levels was observed and recorded.

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AN EXPERIMENTAL STUDY TO DETERMINE THE REDUCTION IN ULTIMATE BENDING MOMENT OF A COMPOSITE PLATE DUE TO AN INTERNAL DELAMINATION

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ABSTRACT

The purpose of this study was to determine experimentally the effects of internal delaminations in a graphite-epoxy composite plate on the plate's ultimate bending moment. The experiments were conducted using 4-inch by 7-inch specimens with a balanced 0 ±45 90 , 8-ply layup. The delaminations were created by inserting a thin teflon disc between two lamina during layup preparation. The location of the disc, i.e. delamination, was varied in each test, and two disc sizes were considered. The test results revealed that delaminations located near an outer surface resulted in a greater reduction in the ultimate moment than those located near the center of the layup. Furthermore, the reduction in ultimate bending moment was found to be independent of the disc size. The tendency for the internal delaminations to propagate at relatively low load levels was observed and recorded.

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I. INTRODUCTION

The recent increase* in the use of laminated, fibrous composite materials, such as graphite-epoxy and boron-epoxy composites, for major structural components on military aircraft has made necessary the thorough examination of these advanced materials in all possible modes of failure. For example, the evaluation of damage in graphite-epoxy composites has become an area of particular concern for the Navy, since the designs of both the F/A-18 Hornet, the Navy's new lightweight fighter/attack aircraft, and the AV-8B advanced Harrier make extensive use of graphite-epoxy composite materials.

A significant amount of work has been done, and is being done, considering impact damage to advanced composite structures. References 3 and 10 are examples of these studies. The primary failures evaluated in these studies were cracks or penetrations normal to the plane of the composite lamina. However, due to the layered nature of the composite material, delayed failure resulting either directly or indirectly from internal delaminations between the laminae, caused by manufacturing defects, low-velocity surface

*The rapid increase in the use of advanced composites is due primarily to their high strength-to-weight ratios, making them appear ideally suited for aerospace vehicle construction.

impacts, fatigue, or battle damage, is also a critical area of concern. For example, survivability tests on the F/A-18 wingbox, conducted by The Naval Weapons Center, Code 3183, China Lake, California, provided data representative of combat damage, Ref 3. A post test inspection, by the author, of the damaged wing skins, using ultrasonic test methods revealed internal delaminations some distance from the actual area visibly damaged. The exact extent of the delaminated area could not accurately be determined with the available test gear; however, considerable delamination was noted. Data regarding this type of failure, i.e. failure due to in-plane delamination, is almost non-existent.

One problem with composite materials at the present time is that *no realistic inspection capability* exists to periodically inspect composite structures for internal flaws or delaminations that is compatible with either shipboard or even land-based squadron environments. Recent studies have demonstrated the potentially effective use of heat crystal coatings, holographic, and ultrasonic techniques for the inspection of composite materials, Ref 13. Although none of these techniques appears practical at this time, the ultrasonic testing procedure being investigated by McDonnell Douglas and Northrop does seem to offer the most promise.

Since the possibility of internal delaminations in composite materials does exist, it was felt that their effect on the ultimate structural strength of the composite materials, and consequently their effect on aircraft survivability/maintainability, should be investigated. Therefore, the objectives of this effort were: 1. To

determine the amount of reduction in the ultimate bending moment in a composite plate due to an internal delamination of varying size and location, and 2. To determine if an existing delamination propagates under loading prior to failure.

II. EXPERIMENTAL PROCEDURE

The material used during the experimental testing was a graphite fiber/epoxy, type AS/3501-6, manufactured by Hercules, Incorporated. This is the same material selected for use in the production of the F/A-18 aircraft. The material characteristics are listed in Table I for reference:

I.	Prepreg Physical Properties		
	Resin Flow, %	12	
	Volatiles, %	0.7	
	Tack	Conforms	
II.	Laminate Mechanical Properties	Panel No.	Test Value Average/Minimum
	0 Tensile Strength, RT, ksi	3796	235/228
	0 Tensile Modulus, RT, msi	3796	20.5/20.2
	Short Beam Shear, RT, ksi	3797	18.1/17.6
	Short Beam Shear, 250 F, ksi	3797	14.0/13.7
	Short Beam Shear, 250 F, ksi (24 hr. water boil)	3797	9.9/9.8
III.	Panel Physical Properties		
	Panel No.	3796	3797
	Fiber Volume, %	64.4	64.3
	Resin Content, %	27.85	27.94
	Density (lb/in^3)	0.0582	0.0581
	Void Content, %	0.15	0.19
	Ply Thickness, Inches	0.0104	0.0105

TABLE I

GRAPHITE-EPOXY MATERIAL PROPERTIES

A balanced 0 ±45, 90 layup for the test specimens was selected since this is similar to the layup proposed for use in the horizontal tail surfaces of the F/A-18 aircraft. The representative 8-ply thickness was adopted due to the limited production capabilities of the laboratory facilities and the desire to insure a balanced layup scheme. Figure 1 is a diagram of the layup scheme. The layup was processed in accordance with the procedures outlined in Reference 7.

The internal delaminations were created by the insertion of an extremely thin non-porous teflon membrane between the laminae during the preparation of each specimen as depicted in Figure 1.* All delaminations were circular, so as not to bias any propagation of the delaminations in any particular direction, and either 1/2-inch or 1-inch in diameter.

Each composite plate prepared was originally 16 inches square. It was then cut using a diamond edged circular saw into six test specimens. Each specimen was rectangular in shape, 4 inches in width, and 7 inches in length. A typical delamination was located in the plate as depicted in Figure 1.

The equipment used to manufacture the composite test specimens is shown in Figure 2. This equipment was capable of automatically processing the composite plate once the initial layup was made. The initial pressurizing and heating process was done automatically using a network employing a press, timers and a thermocouple. Once the

*The author was gratified to discover that essentially the same method of inserting an internal delamination was used by the Grumman Aerospace Corporation in their study of the effects of internal flaws to damage tolerance in the B-1 horizontal stabilizer, Reference 14.

initial curing cycle was complete, the composite was removed from the press and placed into an oven for post curing. All production temperatures and pressures were taken directly from Ref 7 and are reproduced in Table II for reference:

AS/3501-6 CURE CYCLE

1. Place vacuum bagged layup in autoclave.
2. Apply minimum vacuum of 25 inches of Hg.
3. At a rate of 3 to 5° F/min, raise the laminate temperature to $350 \pm 5^\circ$F. During the heat-up, apply 85 + 10, -0 psi when the laminate temperature reaches $275 \pm 5^\circ$F.
4. Hold at 25 inches HG (min), 85 + 10, -0 psi, and $350 \pm 5^\circ$F for 120 ± 5 min.
5. Cool to a maximum of 150° F in 40 minutes minimum time.
6. Release autoclave vacuum and pressure.
7. Remove layup from autoclave.
8. Postcure laminate for 2 hours at 400° F or 8 hours at 350° F in air-circulating oven.

TABLE II

The actual testing of each composite plate was done using a Riehle 300,000 pound test machine shown in Figure 3, with the following modifications.

Each 4-inch by 7-inch plate was supported at two ends and loaded as shown in Figure 4 using a stirrup type device. This device insured an even loading across the specimen. For accurate measurements of the loads experienced during the tests, a wheatstone bridge arrangement was used to provide an input to a Hewlett-Packard 7100 B Strip Chart Recorder.

A total of 18 specimens was tested. Two specimens without delaminations were tested to establish a base for comparison with the samples containing built-in flaws. One specimen was tested with a cleanly drilled hole 1/2-inch in diameter, and another had a hole 3/4-inch in diameter. Seven specimens were tested with a 1/2-inch delamination in varying depths through the composite, and an additional seven specimens had a 1-inch delamination similarly positioned.

Each specimen was placed on the Riehle Machine and loaded in the center as depicted in Figure 6. Each specimen was slowly loaded to failure. The onset of any propagation of the delamination through the specimen was also noted.

Each failed specimen was examined after the tests in order to evaluate its mode of failure and to attempt to detect any general failure characteristics that were common to all of the failed specimens.

Some major restrictions imposed during the experimental procedures should be noted. Throughout these tests all specimens were loaded only once with a steady increase until total failure occurred. This procedure eliminated the consideration of cyclicly applied loads and

the opportunity to evaluate the effect of fatigue on the specimens. Another obvious restriction placed on these experiments is in the type of loading used. A simply supported plate loaded in one-dimensional bending was chosen from the many possible, and some much more complicated, loading schemes. This was done due to the lack of data on any in-plane delamination induced failures, and to the simplicity of the selected procedure. Another major restriction was the choice of an 8-ply balanced layup for the graphite-epoxy composite specimens. Obviously, this choice had a major impact on the test results due to the highly directional properties of any composite lamina.

III. EXPERIMENTAL RESULTS AND DISCUSSION

A. SPECIMENS WITHOUT DELAMINATIONS

The two composite specimens without flaws were tested to provide a basis for comparison with the flawed specimens. The results are presented in tabular form in Section V of this report. As noted earlier, six test specimens were cut from each 16-by-16-inch plate prepared. To check for repetitive characteristics and the quality of the layups being prepared, one non-flawed test specimen was included in each of two separate sixten-inch layups. Examination of the ultimate loads from these two tests given in Section V reveals a deviation of less than one percent. These loads also compare to within two percent of the loads from identical tests conducted by LT R. Freedman, Reference 6. Therefore, an accurate basis for comparison with the flawed specimens had been determined.

B. SPECIMENS WITH 1/2-INCH AND 3/4-INCH DIAMETER HOLES

The ultimate loads determined for the two specimens with pre-drilled holes are given in Section V. An interesting comparison of those results can be made with a simple analysis of the ultimate loads of the specimen based upon the reduced cross-section due to the drilled holes, i.e. a 12.5% and 18.75% reduction in ultimate load

due to the 1/2-inch and 3/4-inch diameter holes, respectively, for the 4-inch specimens. This comparison is significant since some proposed repair techniques for composite skins require the removal of the damaged area as the first step in the repair process.

Examination of the results given in Section V for the reduction in ultimate load reveals that the strength loss caused by drilling a hole through the plate is much greater than the reduction based solely on the reduction in cross-sectional area of the plate, i.e., the cross-sectional area loss with the 1/2-inch diameter hole is 12.5 percent whereas the ultimate load is reduced by 22.1 percent, or almost double that due to the area loss. The cross-sectional area loss with the 3/4-inch diameter hole is 18.75 percent, whereas the tested plate resulted in the ultimate load being reduced by 31.73 percent, again almost double.

As the load was slowly increased from zero, fracture of the fibers in the top layer occurred when the load reached approximately 80 percent of the ultimate load. The fractures were located at the edge of the hole as shown in Figure 7, where classical elasticity analysis for isotropic materials predicts the highest stress concentrations. No delaminations were observed during the loading.

C. SPECIMENS WITH INTERNAL DELAMINATIONS

The data reflecting the reduction of the load carrying capabilities of the composite test specimens due to the internal delaminations is presented in Section V. The correlation between the reductions in the ultimate loads which resulted from the 1/2-inch delaminations and

those caused by the 1-inch delaminations, can be seen in Figure 8. In this figure the position of the delamination is plotted versus the load at failure divided by the ultimate load of the undamaged specimen. Examination of Figure 8 reveals that a delamination located nearer one of the two surfaces tends to have more effect than one located closer to the center of the layup. Indeed, the test results of the specimens with the centrally located delaminations showed only a very slight reduction in ultimate strength.

Another area of concern was the onset of the propagation of the delamination beyond the original built-in, internal defect. This task proved extremely difficult in most cases, since the tests consisted of only one loading cycle, and a detailed inspection of the specimen during the testing was not possible. However, in some cases, especially when the internal delamination was near an outer surface, the onset of the delamination propagation could be detected visually. The propagation of the delaminations was also found to be characterized by cracking noises, probably caused by the breakdown of the epoxy matrix. These noises occurred even when no outward buckling or cracking was visually apparent on the outer surfaces. The onset of the delamination propagation was noted as closely as possible and recorded. The load at which onset occurred is plotted as a function of flaw position and is presented in Figure 9. It is evident from Figure 9 that the onset of the propagation of the delamination occurs at a much lower load level when the delamination is located in the portion of the specimen that experiences a compressive load. The data scatter which is apparent in Figure 9 is attributable to the difficulty in establishing the exact onset of the propagation.

Prior to conducting the experiments it was anticipated that the propagation of an internal delamination, and the ultimate failure of the test specimen, would occur at lower load levels when the delamination was located in the compression area of the test specimen. These results were expected primarily due to an anticipated buckling of the outer lamina. Although buckling was observed when the delamination was in this area, i.e. position 1, the difference between the ultimate loads of the specimens containing delaminations in Position 1 and those containing delaminations in Position 7 are minimal, as can be seen in Figure 8. However, the characteristics of the failure process were markedly different. As can be seen in Figure 9, the onset of the propagation of the delaminations occurred at a much lower load level when the flaw was located in the positions under compressive loading. Another interesting characteristic of the failure process was that when the delamination was located near the surface in tension, the failure occurred catastrophically. When the delamination was located between the lamina in compression, the failure occurred in steps, with one lamina failing at a time until the ultimate failure occurred. These failure processes can be seen graphically in Figure 5.

The one common characteristic of all the tests conducted with the specimens containing the internal delaminations was that the propagation of the delamination was directionally dependent on the graphite fiber orientation in the adjacent lamina nearest an outer surface of the specimen. For example, a delamination located in Position 1 or 7 propagated in the zero degree orientation. An example of a delamination propagation caused by a flaw in Position 2 is

presented in Figure 10. This figure indicates that although the initial propagation of the delamination was in the direction of its adjacent lamina fibers, i.e. 45° , the subsequent delamination occurred in the direction of the fibers in the outer lamina, i.e. 0°.

IV. CONCLUSIONS

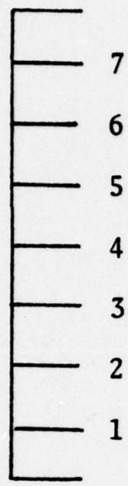
These experiments produced four major conclusions: 1. A very slight reduction in ultimate bending moment resulted from internal delaminations near the center of the composite layup. 2. Relatively large reductions in ultimate bending moments resulted from internal delaminations near the upper and lower surfaces of the composite layup. 3. Propagation of internal delaminations can occur at very low load levels and 4. The reduction in ultimate bending moment appears to be insensitive to initial delamination size.

It is the author's opinion that the relatively large reduction of strength resulting from delaminations nearer an outer surface needs to be examined more closely, since graphite-epoxy layups are being used for flight critical surfaces in the advanced AV-8B harrier and the F/A-18 aircraft.

Another critical area of concern should be that of fatigue failures involving internal delaminations. The experimental results indicate that the propagation of an internal delamination may occur at load levels as low as 20 percent of the ultimate load, well within normal design load limitations. This could seriously reduce aircraft survivability/vulnerability characteristics should these delaminations go undetected. Thus, delamination propagation resulting in the degradation of structural strength, due to operational loading, becomes a distinct possibility. Obviously, not all flight profiles

require loading an aircraft to operational or design limits. However, the loads experienced may be of sufficient magnitude to propagate an internal delamination to the point where loading the aircraft to operational limits could cause catastrophic failure of the damaged structure.

V. RECORDED AND REDUCED DATA



Delamination Positions

A. FAILURE LOADS WITHOUT DEFECT

were 615 and 620 pounds.

B. SPECIMENS WITH 1/2-INCH AND 3/4-INCH DIAMETER DRILLED HOLES

Hole Size	Cross-Sectional Area (%)	Failure Load(#)	Reduction(%)
1/2" Diameter	87.5	483	22.1
3/4" Diameter	81.25	423	31.8

C. SPECIMENS WITH 1/2-INCH DELAMINATIONS

Delamination Position	Failure Load(#)	Strength Lost(%)
1	540	12.9
2	620	0
3	580	6.5
4	610	1.6
5	610	1.6
6	610	1.6
7	555	10.5

D. SPECIMENS WITH 1-INCH DELAMINATIONS

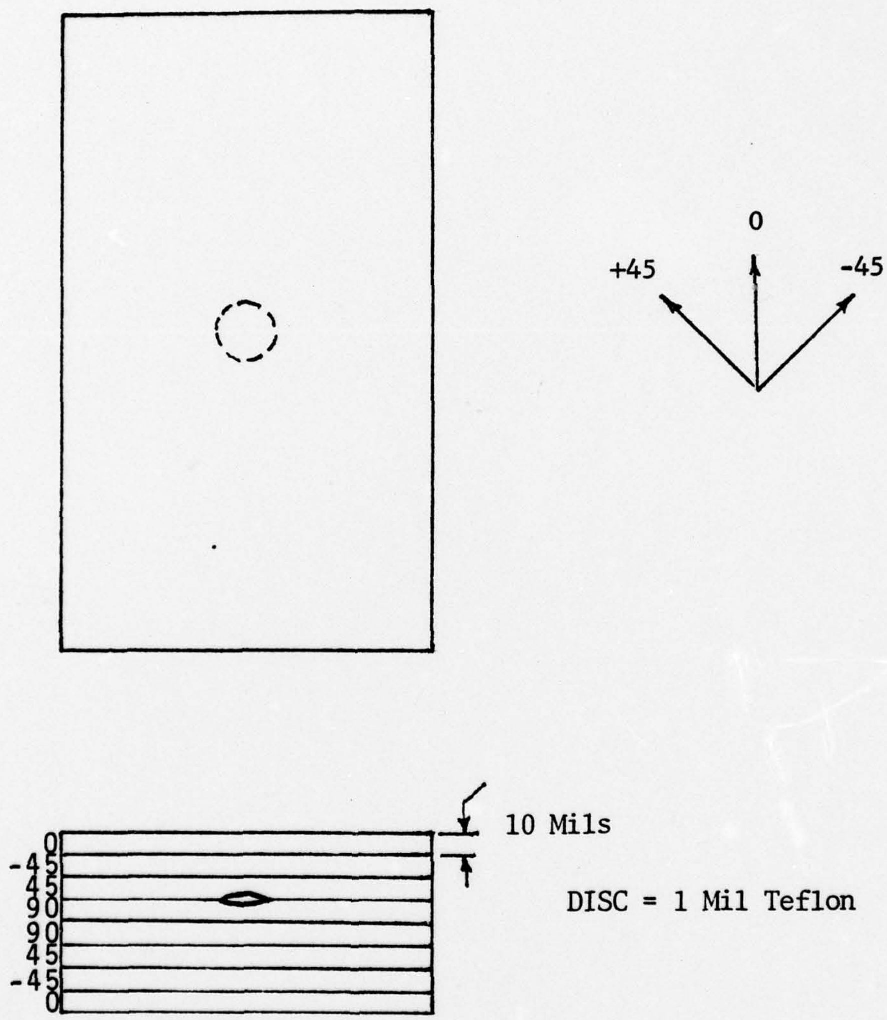
Delamination Position	Failure Load(#)	Strength Lost(%)
1	548	11.6
2	423	31.8
3	625	0
4	620	0
5	625	0
6	605	2.4
7	572	7.7

E. INITIAL PROPAGATION OF THE DELAMINATION FOR 1/2-INCH DELAMINATIONS

Delamination Position	Load at Onset(#)	% of Ultimate Load
1	250	40.3
2	560	90.3
3	390	62.9
4	510	82.3
5	560	90.3
6	590	95.2
7	250	40.3

F. INITIAL PROPAGATION OF THE DELAMINATION FOR 1-INCH DELAMINATIONS

Delamination Position	Load at Onset(#)	% of Ultimate Load
1	120	19.4
2	150	24.2
3	230	37
4	500	80.6
5	625	100
6	605	97.6
7	572	92.3



NOTE: Not Drawn to Scale

Fig 1 - Typical Layup Configuration

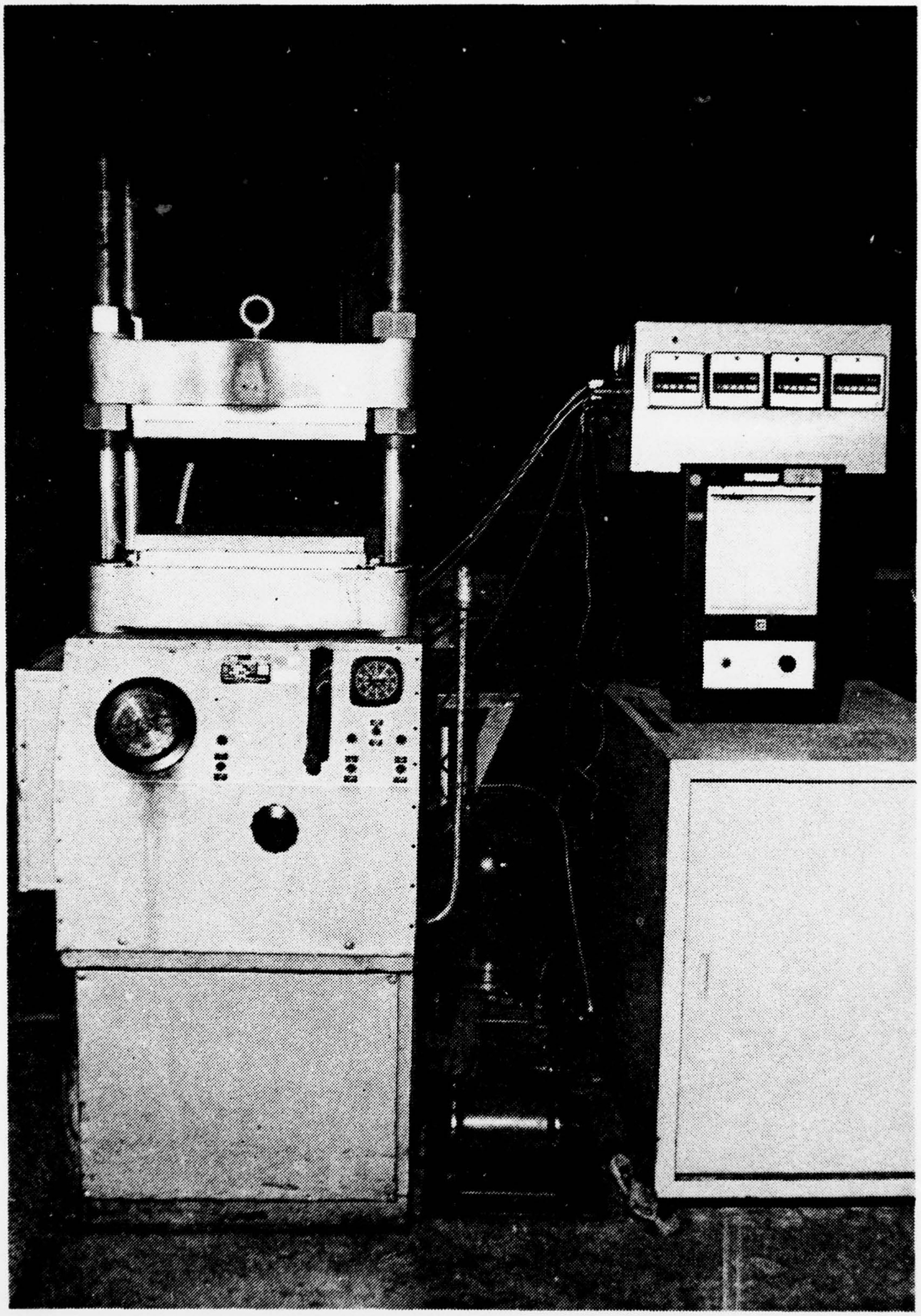


Fig 2 - Processing Equipment

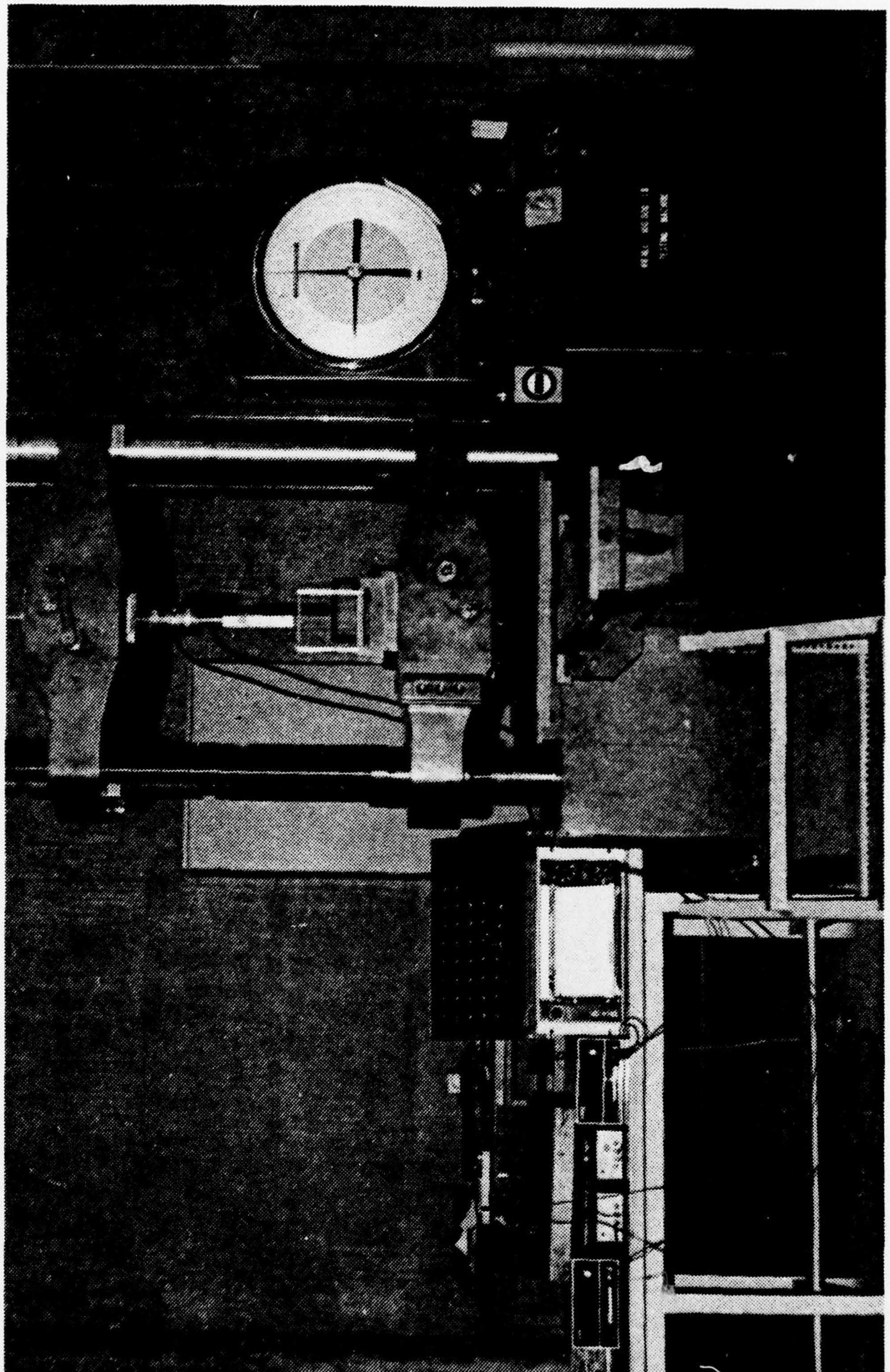


Fig 3 - Load Test Gear

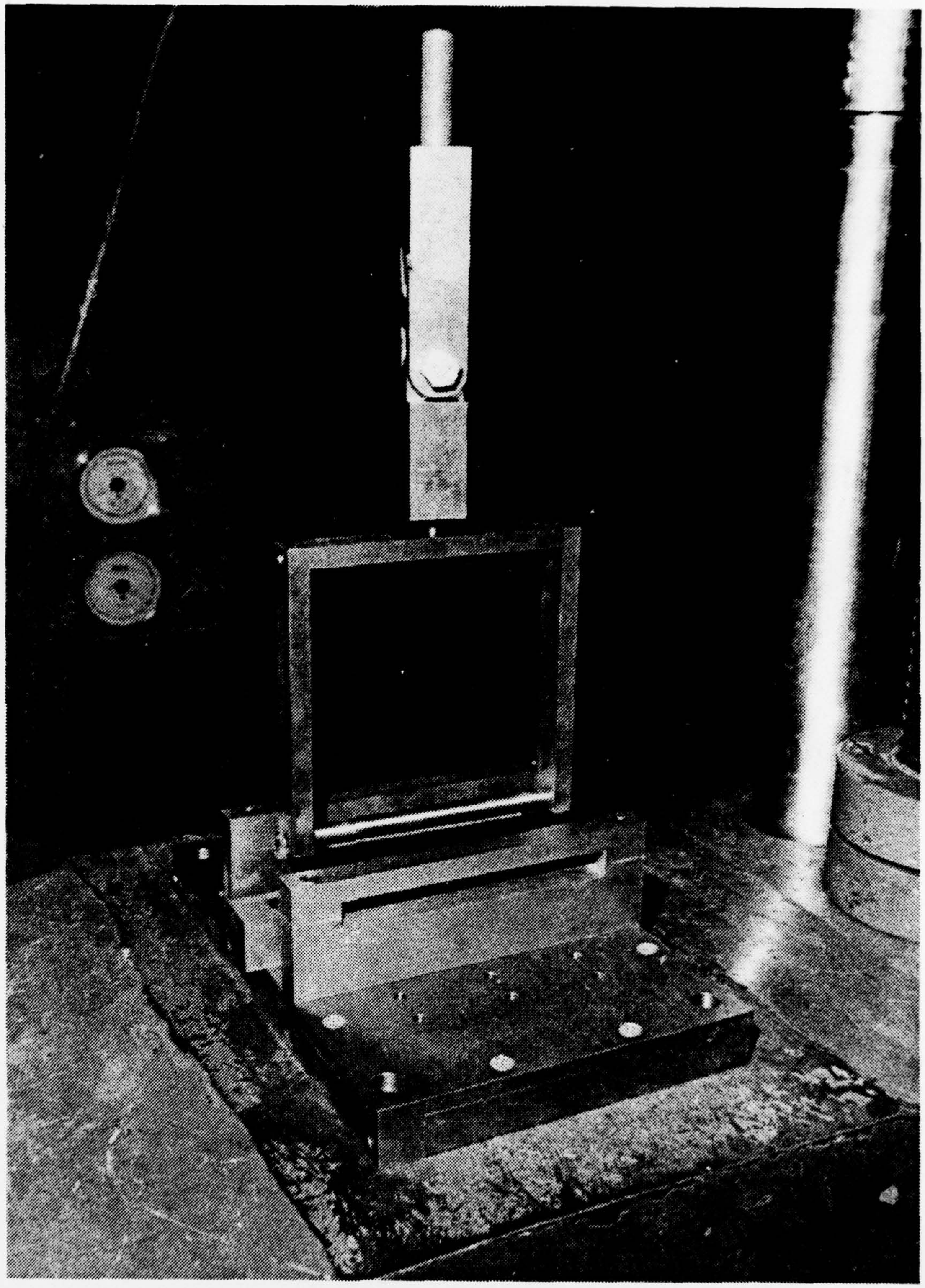


Fig 4 - Loading Apparatus Showing Stirrup Device

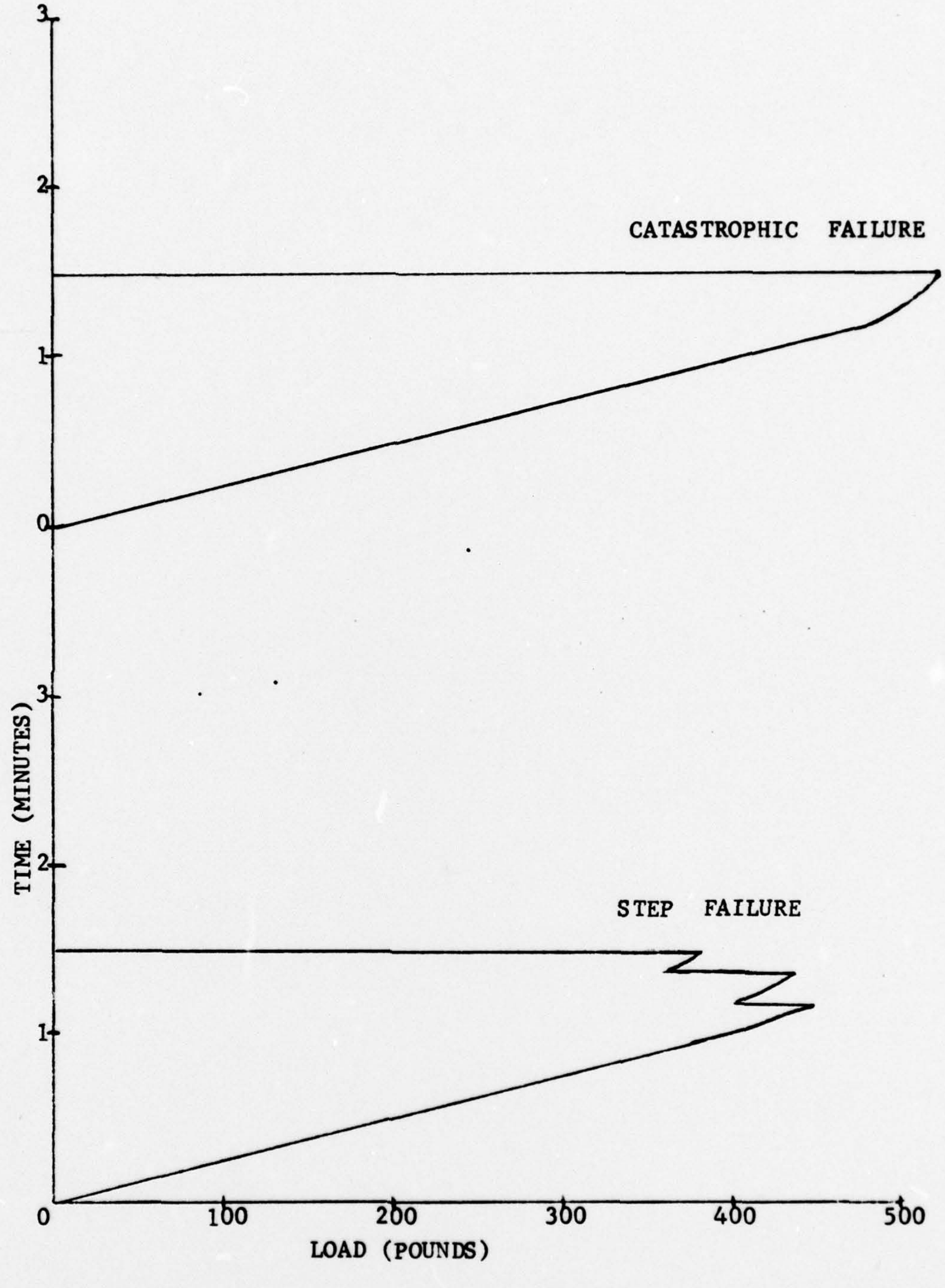


Fig 5 - Specimen Loading Until Failure

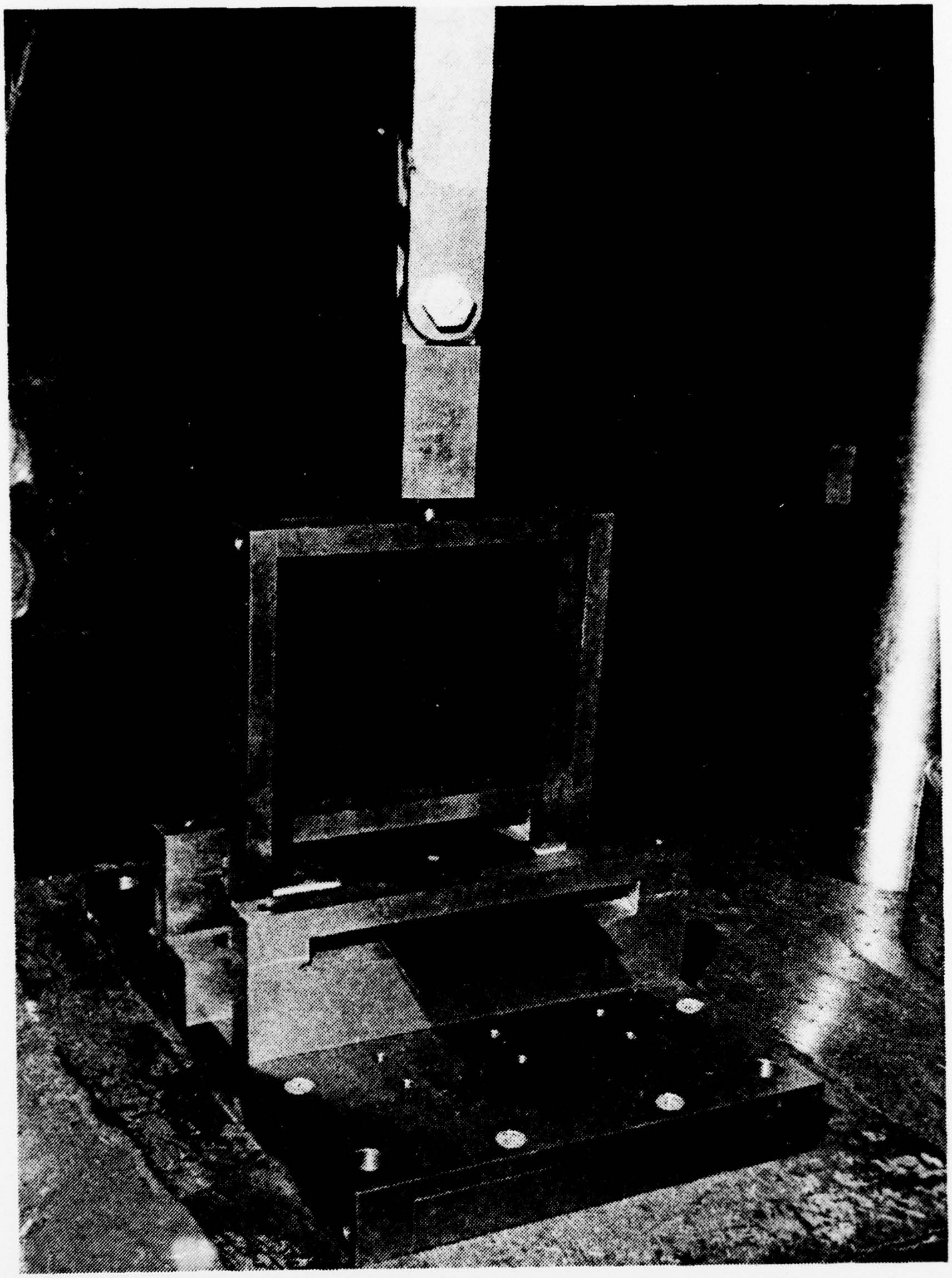


Fig 6 - Specimen Under Loading

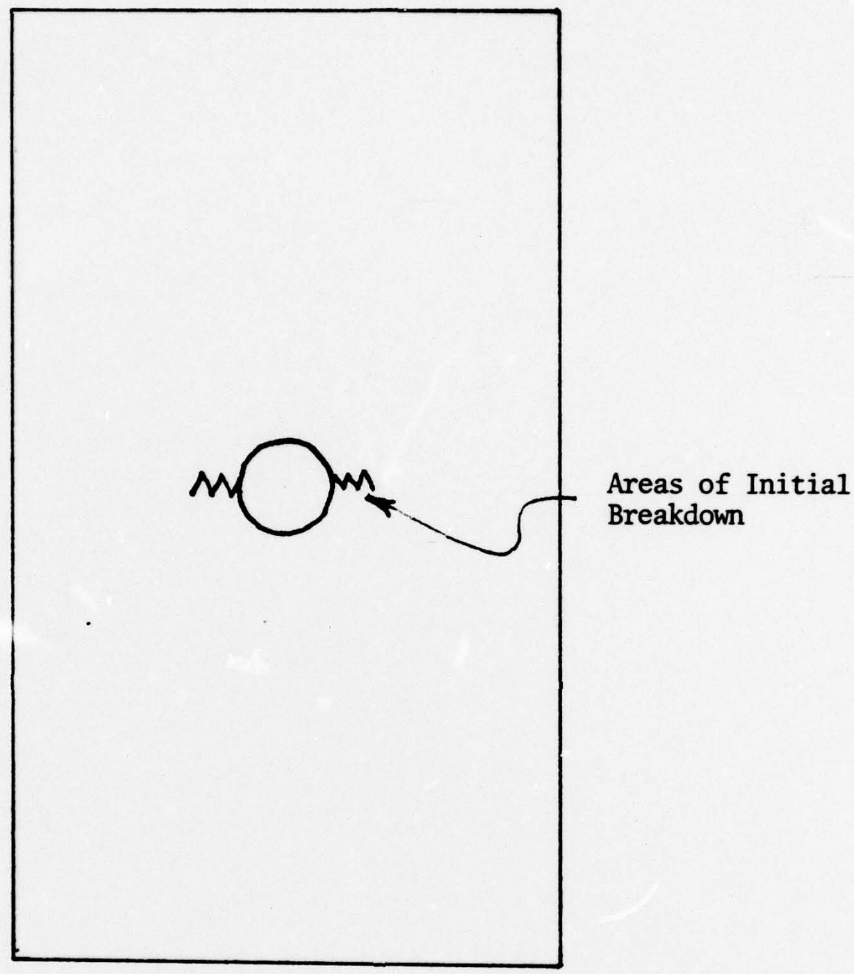


Fig 7 - Initial Breakdown of Specimens with Drilled Holes

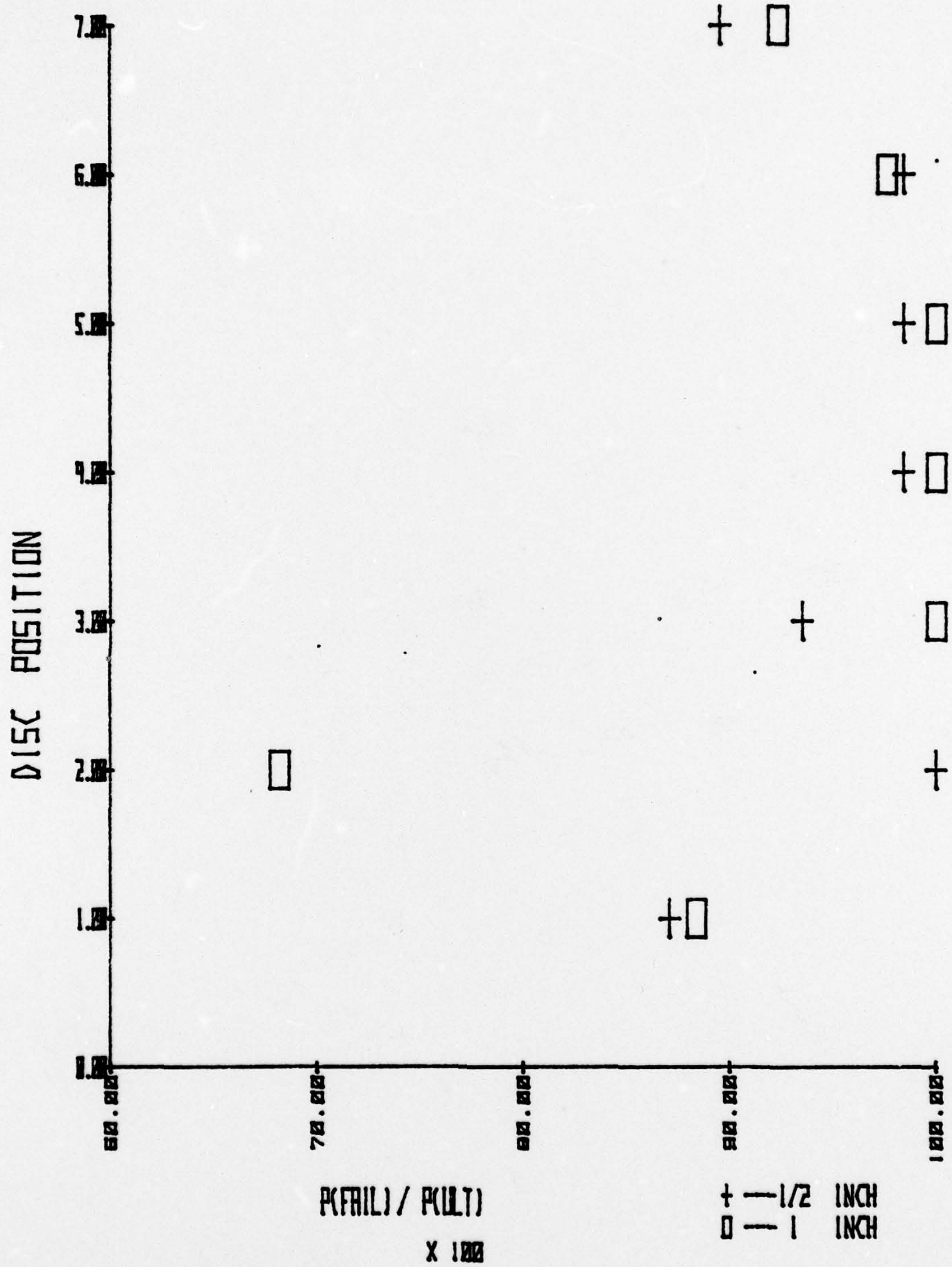


Fig 8 - ULTIMATE LOAD REDUCTION

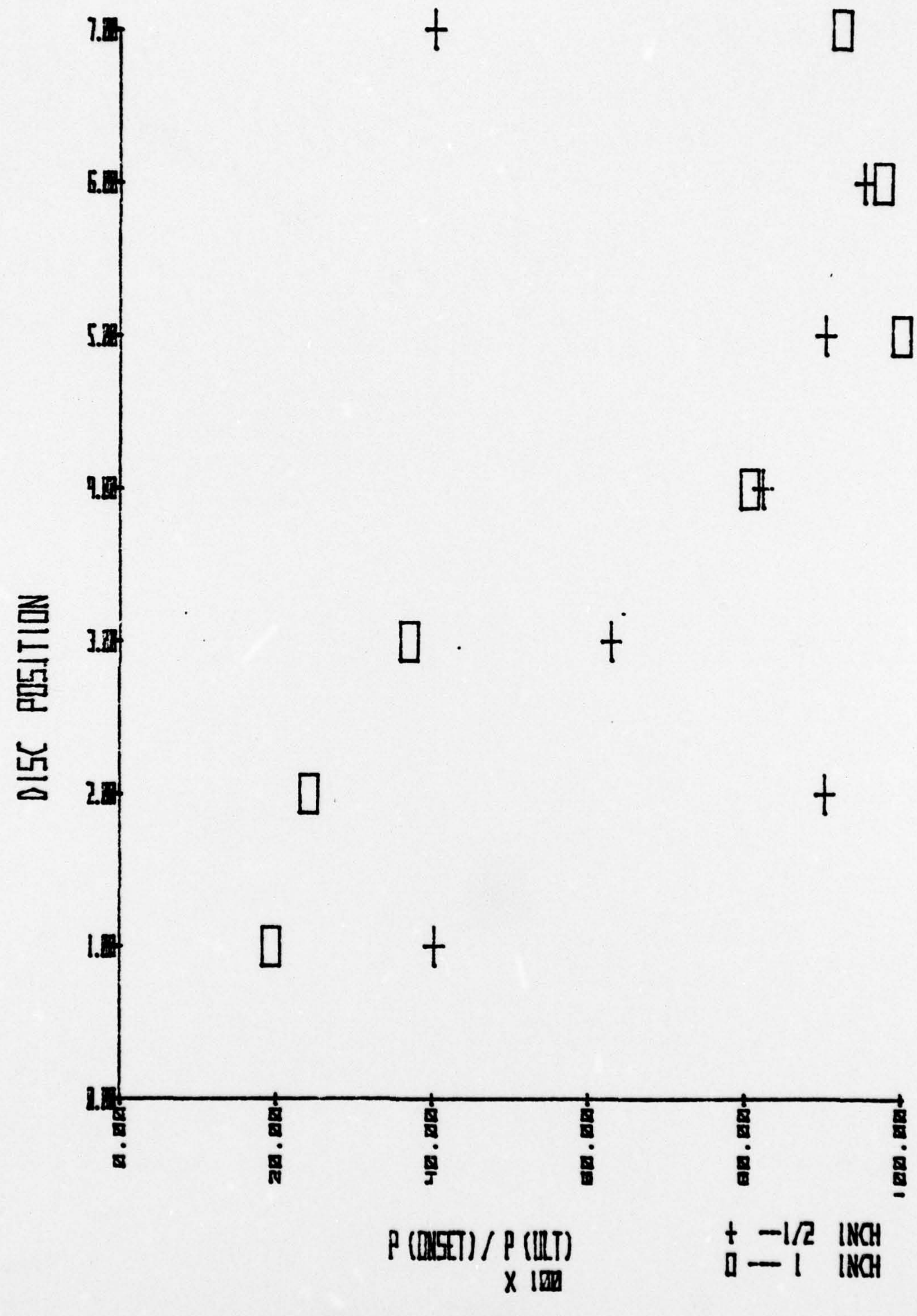


Fig 9 -ONSET OF PROPAGATION

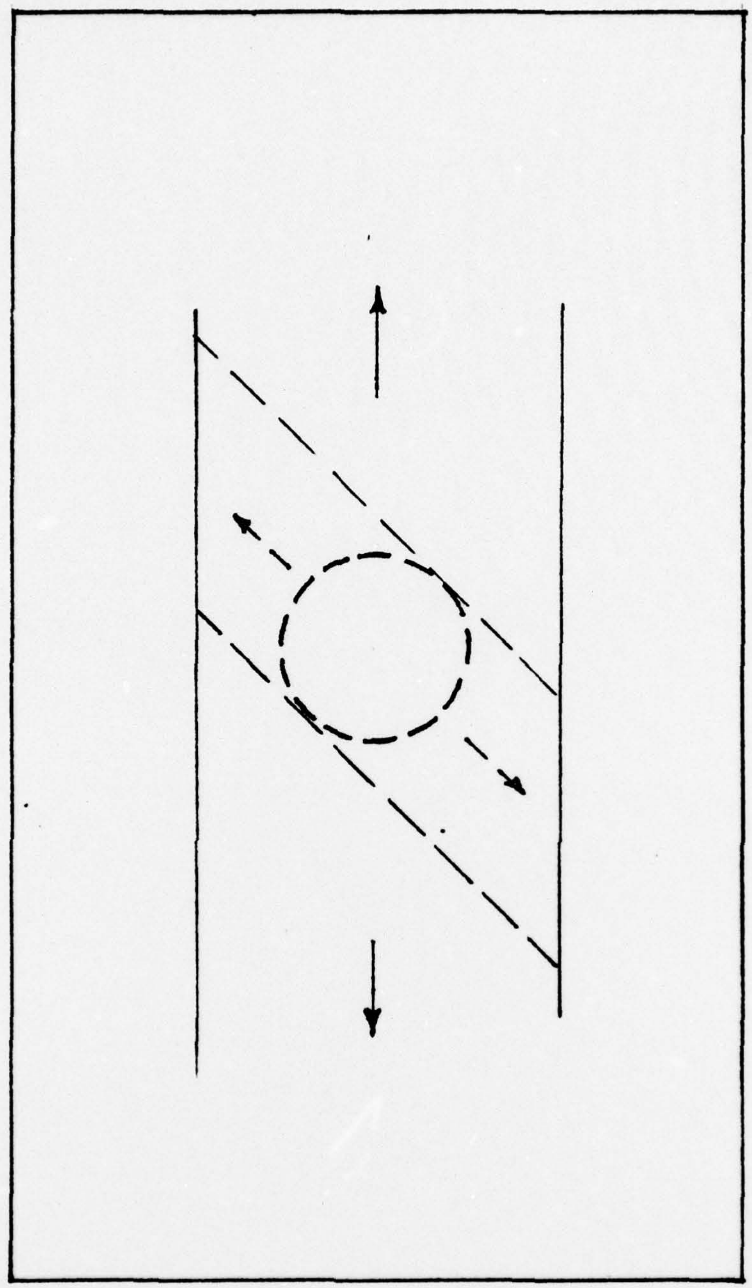


Figure 10 - Propagation Characteristics of Delamination in Position 2

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